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ACTUATOR DESIGN FOR THE SERT II BEAM PROBE

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16. Abstract

The description of the mechanical and electrical design of an actuating device for the SERT II spacecraft is presented. Details of the structural design, lubrication techniques, electrical circuitry, and testing program are given. During vacuum tank testing, one assembly performed 1185 cycles of operation. Two actuators were launched with the SERT II spacecraft on February 3, 1970. One unit accomplished 53 successful operations by June 10, 1970 and then failed during the 54th operation. The second unit accumulated 20 actuations by August 31, 1970, of which 12 were with an operating thruster. Four more were accomplished by October 22, 1970 without an operating thruster. The actuator was performing satisfactorily at that time.

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SUMMARY

The SERT II spacecraft launched on February 3, 1970 had as its main experiment two electrical thrusters. Included in the instrumentation to monitor thruster performance are two hot wire emissive probes. The probes are passed through the thruster exhaust plasma upon ground command by electromechanical actuators. This report describes the mechanical and electrical design of these actuators, and explains the test program which qualified the devices for spaceflight.

Each actuator consists of an aluminum structure in the form of a tapered box beam, a motor subassembly which includes the probe and its supporting arm mounted on the motor shaft, a support and clamp system to hold the probe during launch, and motor control circuitry contained within the structure but as a removable package. This circuitry functions so as to limit the probe travel to a 340° oscillation, with the motor stopping at the end of each arc of travel. A ground command is required to start the motor again. The oscillatory motion permits the wiring to the probe to be continuous. Each probe and actuator assembly weighs 5.65 pounds (2.56 kg), and the actuator power consumption is 10 watts or less.

Two experimental assemblies were built for development testing. The design itself was qualified for flight by a series of vibration, shock, and thermal vacuum tests on a prototype assembly. Five other assemblies were fabricated, two for the prototype spacecraft, two for the flight spacecraft plus one backup unit. These five assemblies were tested prior to installation on a spacecraft by vibration, shock, and thermal vacuum tests but at lower levels than those used in qualification testing. One of the prototype units performed 1185 actuations in a simulated orbital environment.

Two units were launched with the SERT II spacecraft. One unit accomplished 53 operations, but failed during the 54th on June 10, 1970. The second unit had performed 20 operations as of August 31, 1970 of which 12 were with an operating thruster. Four more were accomplished by October 22, 1970 without an operating thruster. It was still functioning well at that time.

INTRODUCTION

As part of the program to develop spaceflight qualified mercury bombardment ion thruster systems, a 6-month continuously operated life test of such a thruster system in a 540-nautical-mile (1 000 000-m) polar orbit has been started (ref. 1). Other objectives of the mission (SERT II) are the validation of ground test results and the determination of ion thruster operating characteristics in the space environment (refs. 2 and 3). Instrumentation on board the SERT II spacecraft for accomplishing these objectives include two hot wire emissive probe systems. These probes are designed to measure the thruster exhaust beam potential and must be capable of 6 months operation in the space environment. It is the nature of the probe filament that it can survive only a few hours continuous immersion in the exhaust beam. Therefore, an actuating device is required which can hold the probe away from the exhaust beam, and then, upon command, sweep the probe through the beam over a specified length of time, and then remove it again. Because of the fragile nature of the probe and filament, the actuator must also provide support for the probe which will permit it to survive the launch environment. The design of this actuator must achieve satisfactory solutions to the problems of adequate structural integrity for the launch environment, electrical systems that perform in ultrahigh vacuum, and the lubrication of moving parts in ultrahigh vacuum and temperature extremes.

This report describes the program which produced the final flight design of this beam probe actuator and produced the flight qualified assemblies. The design effort is discussed in a chronological order with the conceptual design work reviewed first along with the actuator criteria. The actual mechanical and electrical design effort is discussed in the main body of the report. Here the problems of material selection, fabrication and assembly, design for strength, stiffness, and long-term operation in the space environment are discussed. Details of the electrical system are presented along with a discussion of the component selection process. Qualification and testing philosophy and procedures are also discussed with a presentation of test criteria in tabular form.

CONCEPTUAL DESIGN

The first step in the actuator development program was the evolution of a design concept. Two considerations were prominent in arriving at feasible concepts for the configuration and mode of operation. The first consideration was the mounting of the thrusters on a gimbal system such that each thruster can move in a cone angle with an apex of 20°. In order for an actuator to place the probe filament in a repeatable reading position regardless of thruster position, either the actuator must be installed on the gimbal with its thruster, or a unique linkage system must be developed. The former concept was adopted as the more feasible approach and the one which would produce a device with a minimum of moving parts. With the decision to mount each actuator on the gimbal with its respective thruster, the overall size of the actuator was essentially established, a goal of 6.5 pounds (2.94 kg) as the maximum weight was established, the actual actuator-to-spacecraft mounting interface dimensions were set, and the initial specifications of vibration and shock input to the actuator were determined. The original actuator specifications are outlined in appendix A. (See also fig. 1.)

The second consideration was the signal transmission between the probe-transformer system and the associated equipment on the spacecraft. The use of 360° rotation with slip rings for transmitting the signals was considered undesirable because this introduces the problems of reliability of another device plus adds noise to the signal. With partial rotation and the use of a reversing system, the signal transmission could be accomplished with continuous electrical leads. The problems of using a loop in the transformer output cable were accepted as being less severe than the electrical problems associated with slip rings in the space environment.

The configuration for the actuator that resulted from the conceptual design effort is illustrated in figure 2. Note that the probe is shown in two positions; the reading position over the thruster and a launch position. One of the actuator's requirements is to support the probe during launch to enable it to survive the vibration and shock environment. The oscillatory travel of probe and arm permit the location of a permanent support structure between the probe parking positions. This structure, as shown in figure 2, with some clamping means was accepted as the most convenient method of supporting the probe.

During the conceptual stage, three methods of clamping and releasing the probe were considered: (1) subliming solids, (2) pyrofuse wire, and (3) pyrotechnically actuated mechanical device. The subliming solid approach was not pursued because of possible contamination or coating of other experiments. The pyrofuse wire was tried experimentally and the vaporization byproducts proved to be a severe contamination problem. This approach was also terminated. Subsequently, item (3) proved to be the most practical solution.

Also seen in figure 2 is the location of the actuator's electrical circuitry within the structure. This approach saves the weight and space of a separate container and it allows electrical leads between circuitry and motor system to be of minimum length.

Two other electrical decisions were made to complete the conceptual design effort. One decision was to use standard, spaceflight qualified electrical components for the circuitry instead of attempting the development of a printed circuit or integrated circuitry. The standard components were more readily available, and the technicians available were well-trained in working with such components. The second decision was the choice of limit switches to stop the travel of the arm instead of ground command or on-board timer. Qualified switches were available and their reliability appeared to be satisfactory for this purpose.

PRESENTATION OF DESIGN EFFORT

The result of the effort which converted the conceptual design to a flight qualified device is illustrated in figure 3. The photographs show a beam probe actuator assembly complete with probe and transformer, pyrotechnic pin puller, and thermal coating. In these views, the actuator is in the launch configuration with the probe held in the launch clamp by the pin puller. The assembly pictured was the backup unit for the two assemblies launched on the SERT II spacecraft February 3, 1970. The location of the two assemblies on the spacecraft can be seen in figure 4 which shows the "earth" end of the spacecraft in a test configuration.

Figure 3 illustrates that an actuator assembly can be considered as four subassemblies. These four are (1) the structure with the launch support arm, (2) the motor subassembly which is mounted on top of the structure, (3) the launch clamp mechanism fastened at the end of the launch support arm, and (4) the electrical package which is installed inside of the structure. Early in the design phase, it became apparent that considering the actuator as a series of subassemblies would facilitate the design and fabrication. Two obvious advantages result from the subassembly concept. The first is the greater utilization of manpower by performing simultaneous fabrication and assembly operations. A second is that inspections and electrical testing of subassemblies locates and resolves many problems prior to final assembly. The final inspection and test problems are greatly reduced.

Another benefit was gained when the thermal coatings were applied. The black paint used requires a 14-day cure time after application. While the structures were curing, assembly of other items proceeded.

The description of the mechanical and electrical design which follows is divided along subassembly lines.

Structural Subassembly

The support structure of the actuator (see fig. 5) is an aircraft-type box beam of rectangular cross section at its base and tapering to a square cross section at the top. The necessity of locating the motor and its electrical system within the structure require that the structure be hollow and without internal bracing. A box beam is an excellent means of achieving rigidity and lightweight under these circumstances. Aluminum alloy 6061 was chosen as the material. The four sides of the box were formed from 0.032-inch- (0.081-cm-) thick sheet in the T-0 condition (fully annealed) then heat treated to the T-6 condition (full hard) before assembly. Three of the sides have raised darts (large dimpled areas) formed by hydraulic pressure and a female die. The darts provide local stability for the flat areas. The base of the structure is machined from T-6 condition plate and is provided with four 1/4-20 tapped holes for mounting the actuator assembly on the gimbal system. Ribs machined into the area between the mounting holes provide stiffening. At the top of the structure is a flange which adds to the structural integrity of the box and provides a mounting surface for the motor subassembly. The flange is machined in four sections and riveted in place. Aluminum rivets of alloy 2117-T4 condition were used through the structure. Rivets were used because they are lighter than screws and nuts and are easier to inspect than spot welding.

As the actuator is essentially a cantilever beam, the point of maximum stress in the structure occurs in the joint between the sheet metal box and the machined base. The final arrangement of this joint was to bend each box side inward so it would be flat against the base. An interior doubler 0.06 inch (0.15 cm) thick was riveted to each side and bolted to the base along with the side sheet metal This joint is illustrated in figure 5. In addition, the 0.125-inch- (0.31-cm-) thick flange for supporting the electrical package was extended down to the base and bolted to it. At the opposite side, two 0.125-inch- (0.31-cm-) thick stiffeners were added in a similar manner. These are not shown in figure 5, but they are seen in figure 3(b).

An item which is part of the structural subassembly but not a part of the box beam is the structural member, the launch support arm, which supports the probe clamping mechanism and the probe during launch. This part is machined from bar stock because the complex shape and relatively close tolerances required made sheet metal fabrication difficult. Again 6061-T6 aluminum alloy is used. The wall thickness of this part is limited to 0.070 inch (0.17 cm) except at the two ends where 0.12-inch- (0.3-cm-) thick walls are used for fastening purposes; therefore, the weight of the finished part is essentially the same as that of a fabricated sheet metal part. This part can be seen in figure 3(b).

A basic requirement of all electrical packages on the SERT II spacecraft was the provision for venting outgassing products. For the probe actuator, this is accomplished with a 0.75-inch- (1.9-cm-) diameter hole in the box beam. The hole is covered with 100 by 100 mesh of 0.0045-inch- (0.014-cm-) diameter 304 stainless-steel wire. The purpose of the mesh is to permit rapid outflow of materials when the box interior pressure is higher than the external pressure but limit inflow of gas molecules when both pressures have essentially equalized. The vent hole is located as shown in figure 5. The launch support arm described previously covers this area, but it has a large opening in its base over the vent hole.

One duty of this structural subassembly is providing the main thermal control surface area for the overall assembly although all actuator surfaces have some form of thermal control coating. The temperature of the actuator is controlled by means of thermal coatings which have the proper absorptivity and emissivity. Flat black acrylic paint and aluminum foil tape were the materials used. The color pattern was evolved as a part of the overall spacecraft thermal design.

Motor Subassembly

At the top of the structure and mounted to the flange is the motor subassembly. This system has three design features: (1) the motor and its mounting system, (2) the motor lubrication technique, and (3) the two limit switches and their installation. Each of these features is discussed separately.

Motor and mounting equipment. - The motor selected is a hysteresis type, Globe P/N SC59A102, operating on a 115-volt, 400-hertz power source and consuming a maximum of 10 watts power. The unit is a gear motor in which the motor rotates at 12 000 rpm and the shaft speed is 1.12 rpm. The vendor made two modifications: (1) the output shaft was lengthened to 1.07 inches (2.72 cm), and (2) G-300 silicone grease was used as the lubricant instead of the vendor's standard grease.

The original power goal had been 3 watts with a dc source; a sealed, brushless, direct current motor had been obtained that met this requirement. However, the operation of this motor was unsatisfactory at low temperatures. The ac hysteresis motor proved to be satisfactory at all required temperatures and was accepted as the motor for the actuators. The 10 watts ac power proved to be no problem since a 115-volt, 400-hertz source was already on the spacecraft to power another component. Since the actuator operation is intermittent and each sweep lasts a maximum of 50 seconds, sharing of the power source was not difficult to schedule.

The motor is mounted in a container which has a seal on the motor shaft. Figure 6 shows the enclosure is in two parts: (1) the seal housing to which the motor is installed and (2) the container itself. The two parts have an O-ring seal between them, and the opening in the end of the container is also O-ring sealed. The electrical feedthroughs are held in place with an epoxy compound; therefore, the enclosure is hermetically sealed except at the shaft. No attempt was made to design a hermetic seal for the shaft. A close tolerance annular seal was designed instead. The seal is an acetal resin plastic bushing. It is a snug fit in the seal housing and provides a radial clearance of 0.0005 to 0.001 inch (0.0012 to 0.0025 cm) around the motor shaft at normal operating temperatures.

All of the metal parts in the motor subassembly except the seal shim are machined from aluminum alloy 6061-T6. The shim is made from an aluminum laminated shim stock. This material consists of 0.002-inch- (0.005-cm-) thick layers that can be peeled for adjusting the overall thickness of the shim. The two O-rings in the motor container are fluoroelastomer per MIL-SPEC-R-25897.

The electrical feedthroughs are glass to metal terminals, one for each of the three motor leads. The terminals are held in place with an epoxide casting resin chosen because of its -68° to 260° C temperature range, a coefficient of thermal expansion nearly the same as that for aluminum and good electrical properties. It had the strength to resist the pressure forces on the terminals. These forces existed during vacuum tank testing and in the period directly after launch when the enclosure internal pressure was still nearly atmospheric and the external pressure was vacuum.

It should be noted that, in determining the spacing of the three terminals in the container wall, some consideration was given to Paschen's Law for electrical breakdown in vacuum. A calculation was performed, based on air. However, the outgassing of the G-300 creates a gas mixture whose electrical properties at low pressures are unknown. Because of this uncertainty, the terminal spacing used was the maximum possible in the available space.

After the motor is installed within its enclosure, the arm is mounted on the motor shaft. The hole in the arm for the shaft provides a snug fit on it. When the arm is in place, a hollow spring-type roll pin is inserted in a hole through the arm and shaft. Because the roll pin holes were not drilled at assembly, slight differences existed in the diameters of the holes in motor shafts and in the arms. This allowed a few degrees of rotational movement in a plane perpendicular to the shaft in all of the arms but the movement was not a problem.

A groove is milled in the arm as a resting place for the cable between the probe and transformer. In this groove, the cable is protected from the plasma particles when the actuator is functioning. Additional protection is provided by a cover which completely encloses the cable. The cover is not shown in figure 6 but is clearly visible in figure 3. In addition to protecting the cable, the cover is a convenient surface for thermal coatings and does add torsional stiffness to the arm. Also, it covers the roll pin and is added assurance that vibration and shock cannot dislodge the roll pin.

One function of the arm is to actuate the two limit switches that turn off power to the motor at the end of each sweep. Two inclines with a flat surface between is machined on the bottom of each arm. These surfaces act as cams to trip the switches when the arm slides onto the leaf of each switch. In order that excessive wear and/or cold welding be prevented, the arms were given a hard anodize treatment per MIL-A-8625B, Type III, Class 1. This type of anodize is a hard, thick, abrasion-resistant surface ideally suited for preventing sliding wear. As an additional precaution, the leaf of each switch was burnished with molybdenum disulfide powder. An examination of the arm and switches on the actuator that performed 1130 cycles of operation in a simulated space environment revealed that no measurable wear had occurred.

Motor lubrication. - The solution to the problem of lubricating the motor and its gear box in the space environment consists of two parts. They are (1) the use of the G-300 silicone grease and (2) enclosing the gearmotor in a container which provides an annular seal for the shaft. The reason for selecting the annular-type shaft seal is explained in the next paragraphs.

When the actuator is first placed in a vacuum environment with atmospheric pressure inside the enclosure, the flow through the seal is in the viscous regime. The seal is not very effective in this regime and the interior pressure gradually decreases until it reaches the 10^{-4} to 10^{-5} torr level. Here two changes occur. The flow changes to molecular flow, and in this regime, the seal is a severe restriction. Also, the enclosure's interior pressure is now low enough to start the outgassing of the G-300 grease. This grease, because it contains a series of polymers each with its own vapor pressure, does not have one specific vapor pressure. In general, at the actuator's operating temperatures, the resultant vapor pressure will be in the 10^{-6} torr range. As long as the outgassing continues, the effect of the seal will be to keep the interior pressure in the 10^{-6} range, while the exterior pressure is approximately 10^{-12} torr. With these conditions, the mass of the G-300 grease is isolated from the very hard vacuum of space and the outgassing rate is retarded.

It is a characteristic of the G-300 grease, that its outgassing rate and, hence, weight loss rate are not constant with time. Tests conducted by the vendor have shown a weight loss of about 2 percent in the first 24 hours, and then a slightly smaller percentage in each succeeding 24-hour period. These tests were conducted at 65° C and at pressures of 10^{-6} to 10^{-7} torr. The motor-operating temperature is not expected to exceed 50° C. Thus, with this temperature and the pressure maintained within the enclosure, the motor is expected to have adequate lubrication for the 6 months orbital period. During the many vacuum tank tests prior to the launch and during the flight, no evidence of lubrication failure has been found.

Limit switches and their application. - The limit switches are leaf operated, hermetically sealed, SPDT types as specified by MIL-S-8805/46. Two mounting holes are provided in each switch but the largest fastener that can be used is a number 2 machine screw. During prototype vibration testing, some trouble was experienced with one switch that moved from its installed position. It was decided not to rely alone on the number 2 machine screws in future assemblies. Additional fastening was achieved by means of an epoxy compound placed around the switches after they were installed on their brackets. The material selected is a thixotropic epoxide adhesive which has adequate strength to prevent any movement of the switches.

An important consideration in proper application of the limit switches is achieving the correct position of the switches in relation to the cam surfaces on the arms. This is necessary in order that at the end of each sweep, the switch leaf will be depressed the required amount to close the contacts. For the probe actuators, this was achieved by placing adjustable shims under the switch brackets, and then adjusting the shim thickness to place the switch in proper position. This position was selected in the following manner:

(1) The switch bracket was raised until each switch was just actuated with the leaf in contact with the flat surface under the arm. The trip point was determined with an ohmmeter. The switch requiring the greater depression of the leaf was the governing one.

(2) The shim thickness was then increased 0.008 to 0.010 inch (0.02 to 0.025 cm) to insure each leaf would be depressed this amount beyond its trip point.

On certain switches, the tip of the leaf was very nearly in contact with the top of the switch can when this was done. When the arm is in launch position, it is directly over the switch and the leaf is fully depressed. To eliminate the possibility of the leaf tip striking the can during vibration, causing a puncture and thus, a loss of hermetic sealing, the tip of the leaf in these cases was filed flat.

The actual setting of the switch positions was performed after the actuator assembly was completed. This was necessary to compensate for the slight deflections in arm position caused by the probe launch clamp, and the force created by the large loop in the transformer output cable. Also, performing the adjustment as the final item in the assembly process insured a minimum of handling of the actuator between switch adjustment and performance acceptance testing. After the use of the epoxy adhesive and the position setting technique just described were incorporated in the assembly procedures, the settings achieved proved to be completely reliable during all ground testing.

Launch Clamp Subassembly

The last of the mechanical subassemblies is the probe launch clamp. This unit is fastened to the end of the machined arm which is part of the structural subassembly. Figure 7 shows the launch clamp and illustrates that it consists of three main parts: (1) the clamp itself, (2) the bracket which supports the clamp, and (3) the pyrotechnic pin puller. All of the machined parts are made from aluminum alloy 6061-T6. As shown in figure 7, these parts have shims between them. The shims are needed to adjust the clamp's position to match the probe's rest position - a necessary step for reliable release of the probe after launch. The release is accomplished by firing the pyrotechnic charge in the pin puller and this causes the pin to retract within the pin puller body. The probe is then free to move out of the clamp.

The design of the clamping system had to consider the structure of the probe. This item is a thin wall, rectangular cross-sectioned tube which cannot endure much clamping force. The tube can be crushed by an excessive force on its sides. Thus, the probe must be held gently, no matter how firmly it is clamped. The clamping system which resulted from this consideration holds the probe on all four sides. On the two narrow sides of the probe's cross section, the clamp does not exert a force. The gap in the clamp into which the probe fits is designed to be 0.001 to 0.005 inch (0.002 to 0.01 cm) greater than the height of the probe. Thus, the probe is held along the two narrow sides without force but with a minimal clearance.

All of the clamping force is exerted on the probe's long sides. In order to do this and control the amount of force exerted on the probe, two steps were taken. The involved clamp dimensions were given close tolerances to prevent wide variances in the clamping gap. Also, a butyl rubber shim was placed between the probe and the clamp. The dimensions were set such that this shim is compressed to two-thirds of its free thickness when probe and pin puller are in position. Several advantages arise from the use of the shim. The clamping force on each probe-clamp subassembly is essentially the same regardless of small dimensional differences, and this reduced the concern about an excessive force on any one probe. Also, the shim is in shear for motion along the length of the probe-arm combination. Some damping effect is, therefore, provided during vibration in this direction; the Y-axis in figure 7.

For holding the butyl shim in place many adhesives were considered. However, butyl rubber is a difficult material to hold with an adhesive. Rather than risk having the shim break loose from the clamp after the spacecraft was in orbit and drift into the thruster or actuator mechanism, a mechanical retaining system was devised. A small groove was cut into the face of the shim, and a stainless-steel wire was fit into the groove. One end passed through a hole in the clamp, and the two ends were tied together. This method was effective for holding the shims in place throughout all ground testing.

In deciding where to place the clamp and pin puller along the length of the probe-arm combination, the question of what location would have least effect on the system had to be answered. The probe and arm together can be considered a pendulum with the motor shaft as the pivot. If a position for the clamp could be found whose motion input in the Z-axis during vibration would cause rotation and not translation about the motor shaft, the static load on the motor bearings would be reduced. The center of percussion of the probe-arm combination is such a location. Calculations showed the center of percussion to be at a convenient location on the probe body, and this was the location used. An advantage of this choice is that the center of percussion and axis of rotation are interchangeable; thus, motion input from the motor shaft in the Z-axis does not cause translation of the probe. This effect is useful in one axis only, but in the case of the actuator, this one axis has the most severe vibration input to the probe and transformer. Therefore, the advantage is worthwhile.

When the clamp subassembly is bolted to the actuator structure, a butyl rubber shim is placed between them. Experiments with a shim in this location during the early testing showed the effect of the shim to be beneficial. The only problem with the butyl shim was achieving a proper torque on the fasteners and still keep the two metal surfaces parallel. The use of a torque wrench alone did not solve this problem. The gap between the surfaces was measured and adjusted by adjusting the torque on the fasteners. As self-locking fasteners were used, their holding ability was assured after the minimum required torque was achieved.

With the clamp subassembly installed on the structure, the adjustment of the metal shims between the clamp and its bracket can be accomplished. Because of the close tolerance gap in the clamp along the X-axis, the height of the clamp must be set so that, when the probe and arm are resting in a natural position, the probe will be centered in the clamp. The adjustment is accomplished both with solid shims and those which have 0.002-inch (0.005 cm) laminations.

One interesting problem to be solved was how to place the arm in launch position with the probe seated in the clamp. This had to be done prior to all dynamic environmental tests, and whenever a component separately or on a spacecraft was transported any distance. To accomplish the movement of the arm into the launch clamp, a portable power supply and command system was built. This unit supplies 115-volt, 400-hertz power, 28-volt dc power, and a sweep command system. It was connected to the electrical system of the actuator at the connector on the electrical subassembly. The operational procedure was as follows:

(1) The actuator was operated to get the arm traveling in a counterclockwise direction.

(2) Before the arm reached the counterclockwise switch, the leaf of the clockwise switch (see fig. 9) was depressed.

(3) When the arm reached and depressed the counterclockwise switch, it did not stop and remained in motion.

(4) When the arm was over the switch and the probe was in the clamp, the 115-volt, 400-hertz power was switched off.

With this procedure, the probe could be returned to the launch clamp whenever necessary and in a convenient manner. Two factors help this operation. The motor stops almost instantly without coasting when the power is switched off. As previously mentioned, each arm had some slight rotational movement about the motor shaft. Because of these two items, it was possible to place the probe within the clamp without difficulty.

Electrical Subassembly

The last of the units that make up a probe actuator is the electrical subassembly. The close integration of both mechanical and electrical design work was required for the electrical subassembly in order that a compact unit would be achieved. This was made necessary by the limited space available within the actuator structure. Another major problem was achieving a sturdy support system which could keep vibration levels to a safe value.

Figure 8 shows an overall view of the electrical package and the general placement of the components. The electrical schematic is shown in figure 9. This portion of the report will describe only the mechanical design. The operation of the circuit is described in appendix B.

The electrical subassembly does not contain a chassis. Instead, components are mounted on terminal boards which are then fastened within an aluminum frame. The frame is a three-sided structure, fabricated from a single sheet of 6061-T0 alloy. After bending, it is heat treated to T-6 condition and riveted. At the open end of the frame, a terminal board serves as a structural member. This is the same board which supports the motor power-factor capacitors. The board which supports the five relays also serves as a structural member and as an insulator. It is a beam with both ends fixed and has a 0.18-inch (0.45-cm) thickness necessitated by the weight of the relays. All of the other boards are 0.06 inch (0.15 cm) thick, and all are made from epoxy glass sheet, type GEE, per MIL-P-18177.

For the selection of electrical components, a preferred components specification list was used. When a component was purchased by a specification on this list, the vendor was required to supply written certification of compliance with the specification. A receiving inspection on a small percentage of these items was performed. If no specification was available for a component, the items were all subjected to a rigorous receiving inspection.

Each terminal board was wired individually and then installed into the frame at which time the internal connections were made. On the two stacked boards, a fastening system was used which permitted them to be stacked after wiring but prior to installation in the frame. Also, prior to installation, each board was given a conformal coating in accordance with Kennedy Space Flight Center Procedure KSC-E-0001.

One reason for the spacing of the terminal boards as shown in figure 8 was the use of polyimide insulated wire. This wire requires a generous bend radius, and the location of the boards had to consider this requirement. For example, the connecting wires between the relays and the stacked boards form large loops with the bend near the capacitors. All connections within the package are in the form of loops, and the central volume is filled with wires. This was of no concern because of the excellent insulating and abrasion resisting properties of polyimide insulation.

The installation of a completed electrical subassembly into an actuator consists of soldering the lead wires from the motor and the limit switches to appropriate terminals on the subassembly. Then the subassembly can be inserted into the structure for mechanical fastening.

Figure 8 illustrates the support system used. Movement of the two sides is restricted by acetal resin plastic supports installed with shims to provide a snug fit. Also, the base of the electrical package is fastened with screws to the structure. Thus, four sides of the package are supported and vibration tests conducted on packages supported in this manner produced good results. The force levels on components were kept within acceptable values when the dynamic input to the actuator assembly simulated the launch environment.

QUALIFICATION AND TESTING PROGRAM

As stated in the INTRODUCTION, the actuator's design had three main problems to overcome: (1) structural integrity for the launch environment, (2) lubrication of moving parts in the orbital environment, and (3) the operation of electrical systems in the orbital environment. A comprehensive test program was required so that the validity of the design could be verified.

This program was divided into two phases. The first phase, qualification testing, served the purpose of determining the flight worthiness of the design. This test was performed on one assembly designated the qualification model and consisted of vibration, shock, and thermal vacuum tests each at appropriate levels. Each was followed by a performance test to determine that the actuator was still functioning properly. The second phase of testing was the component flight acceptance tests whose purpose was to determine the flight worthiness of each assembly. It was performed on the two units assembled for the prototype spacecraft and the three units assembled for the flight spacecraft. The environmental portion of this test phase is described in appendix C while the performance test is described in appendix D. The tests in appendix D were also used in the qualification phase.

Design Qualification Testing

As the qualification test program was specifically intended to prove the flight worthiness of the design, its test program's environmental phase was more extensive and more intensive than that outlined in appendix C. The important features of the qualification program are as follows:

(1) For the vibration and shock testing, the actuator was mounted on a gimbal system along with a thruster. This complete assembly was installed on a spacecraft structure which served as the fixture. An actual transformer and probe were on the actuator, and a pin puller containing a pyrotechnic charge held the arm in the launch clamp. All of these items were also being qualified in this test.

(2) The input levels to the spacecraft for vibration (but not shock) were greater than those for the prototype and flight spacecraft tests. This resulted in generally higher inputs to the actuator for both sine and random conditions than occurred during the component tests where the fixture was a solid plate.

(3) After the dynamic testing and prior to the thermal vacuum testing, the actuator was subjected to a quick pumpdown which simulated the rapid pressure change of a launch. The assembly was placed in a vacuum tank at ambient pressure and temperature, and then was subjected to a change in pressure from atmospheric to 25 millimeters of mercury in less than 90 seconds. The pumpdown continued to 1×10^{-6} torr pressure at which time the thermal vacuum test was started.

(4) The thermal vacuum qualification test lasted a total of 14 days during which time the actuator was required to perform 250 sweeps. The cold passive period was as stated in appendix C but the hot passive temperature was 71° C. These were followed by a 12-day test at 60° C. Pressure was 5×10^{-6} torr or less during the entire test.

During the various phases of the aforementioned program, the performance test outlined in appendix D was used to determine that the actuator was still functioning properly. In all cases, the results of the environmental and performance tests were satisfactory, so the design was accepted as being flight qualified.

However, further testing was performed which resulted in increased confidence in the design. One actuator on the prototype spacecraft performed 1130 sweeps during a 22-day thermal vacuum test of the spacecraft. At the end of this period, the test was terminated so an inspection of components could be made. The only sign of deterioration on the actuator was the fraying of the insulation on the wire loop that twists. However, the probe's performance had not been affected. The spacecraft was returned to vacuum conditions for another week during which the actuator performed another 55 sweeps. At the end of this test, the actuator was functioning properly.

Further vibration testing resulted when an electrical change became necessary in the transformer design which had been qualified with the actuator. A rerun of the qualification vibration and shock test was made utilizing the same assembly of components used in the first test. Thus, the qualification model of the actuator was subjected to two vibration and shock tests, the second time functioning as a fixture. However, when the second test was completed, a full performance check was made and the results were within the limits of appendix D. The only noticeable effect of the second dynamic test was a slight loosening of the arm on the motor shaft.

One last comment about qualification testing concerns the limit switches. As previously stated, the final design of the switch fastening technique involved the use of an epoxy adhesive. The need for this adhesive was discovered during prototype spacecraft testing which occurred prior to the start of the actuator's qualification program. Thus, the final switch fastening technique could be incorporated on the qualification assembly, and it was tested in this configuration.

Prototype and Flight Component Testing

The remaining comments on testing concern the component testing of the actuators for the prototype and flight spacecrafts. After each assembly was completed and all adjustments finalized, it was first subjected to a performance acceptance test (PAT). If the results of the PAT were satisfactory, the assembly was considered ready for component testing and was started through the program. For the actuators, the first step was to clamp the arm in launch configuration using a dummy pin puller. For all component testing, the fixture was a flat plate to which an actuator was bolted. The plate in turn was fastened to the various test devices as required. Vibration and shock testing came first, followed by a performance test. The dummy pin puller had to be removed prior to the performance test. If the PAT results were again within the prescribed limits of appendix D, the assembly proceeded through the vacuum test. It should be noted that in no case did an actuator fail during a component vibration or shock test. During the thermal vacuum test, each actuator was operationally tested four times with instruments and visual observations used to verify its performance. At the end of the test, and when the assembly was again at ambient pressure and temperature, another PAT was performed. If the results were satisfactory, the assembly had completed all component testing successfully and was ready for installation on a spacecraft. None of the actuators experienced a failure during component thermal vacuum testing.

The component testing described previously was only a part of the overall test program for the SERT II Project. Each assembled spacecraft with all its components onboard was subjected first to electrical integration tests, and then a complete program of environmental testing. This included vibration, shock, and thermal-vacuum testing. When operation of a particular component was necessary or feasible during some phase of the testing, this was done. Thus, the beam probe actuators were operated during the spacecraft thermal vacuum tests. The two failures which occurred on actuators happened during spacecraft testing and were discovered while the actuators were operated on the spacecraft. The first occurred on the prototype spacecraft during vibration testing when the failure of a clamping device on the gimbal system created a high gravity input to one actuator. A limit switch on this assembly was moved from its set position, and the actuator would not function. It was at this point that the use of an epoxy to hold the switches in position was introduced which solved this problem. A second failure occurred on the flight spacecraft when again a limit switch did not perform as required. In this case, the initial switch setting was evidently not adequate. The practice of setting each switch at least 0.008 inch (0.02 cm) beyond its actuation position was instituted and this problem was eliminated.

A tabulation of the many tests performed on the actuators is presented in table I.

CONCLUDING REMARKS

On the evening of February 3, 1970, the SERT II spacecraft was launched into a 540-nautical-mile (1 000 000-m), circular polar orbit. In the period since the launch and June 3, 1970, the actuator which is mounted next to the operating thruster had been operated at regular intervals for a total of 52 sweeps. On June 10, 1970, this actuator was again operated. The clockwise sweep (number 53) was normal, but on the counterclockwise sweep the motor did not stop at the end of the sweep. The most feasible explanation of this failure is that the counterclockwise limit switch had malfunctioned or completely failed. The other actuator had only been operated infrequently until June 3, 1970, in order to check its operational availability. Table II lists all actuator operations achieved by June 3, 1970.

After the first thruster failed, July 23, 1970, the second thruster was placed into operation on July 24, 1970. The second beam probe actuator accomplished eight sweeps with an operating thruster until this second thruster failed on October 17, 1970. Four additional sweeps were made as part of the attempt to restart the second thruster. All sweeps with the second actuator were satisfactory.

Only two indications of actuator operation are returned to the ground station. One is the voltage decay curve which indicates the end of each clockwise sweep. The other is the probe filament output recorded as a voltage-against-time curve on a strip chart. This curve provides a means for observing the sweep times within limits. For both actuators, the time of sweep duration had remained constant, excluding the sweep during which the failure occurred. This indicates that the motor circuitry and lubrication techniques are both functioning as expected. Also, as the probe filament data have been good, both actuators have obviously successfully served the purpose of protecting the probes from the launch dynamic environment. Both the actuator design and the testing program used to qualify the design have been essentially adequate; however, a more stringent acceptance test program for the limit switches might have prevented the one failure that did occur.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, October 29, 1970,
704-13.

APPENDIX A

BEAM PROBE ACTUATOR INITIAL SPECIFICATIONS

Operational specifications are as follows:

(1) 340^{o} per sweep

(2) Time for probe tip in beam, 5^{+2}_{-0} seconds

(3) Required life, 250 sweeps through beam

(4) Signal to indicate end of clockwise sweep

Restraints are as follows:

(1) Probe position relative to ion beam specified

(2) Maximum power consumption goal, 3 watts (later amended to 10 W)

(3) Maximum weight goal, 6.5 pounds (2.94 kg)

(4) Interface with probe and transformer (see fig. 1)

(5) Interface with allotted mounting surface on spacecraft

Environmental design specifications are given in the following table:

Vibration		
Sinusoidal (three axes mutually perpendicular)		
cps	g's (0 to peak)	Sweep rate, octane/min
5 to 19	(a)	1.0
20 to 2000	9.0	
Random (three axes mutually perpendicular)		
cps	g's rms	min/axis
20 to 400	6.5	9
400 to 2000	18.9	

[a]0.5-in. (1.27-cm) double amplitude.

APPENDIX B

ELECTRICAL OPERATION OF THE SERT II PROBE ACTUATOR

At the start of the actuator's operation, the following situations must exist:

(1) The latching relay K3 must be set as shown in figure 9.

(2) The counterclockwise limit switch must be held closed by the arm which is still in the launch clamp.

(3) The pyrotechnic pin puller must be fired so the arm is free to move.

The first command signal energizes K1 coil which closes the K1 contacts and energizes K5 coil and K2 through the normally closed contact of K4. Current flow through the closed counterclockwise switch energizes the K3 coil and also K4. This deenergizes K2 but power is not interrupted as all these operations occur within the 50-millisecond command pulse duration. When K5 was energized, the motor started rotating and it continues until the arm leaves the limit switch. Then K4 is deenergized, K5 is deenergized and power is removed from the motor. The actuator now is ready for sweep operation.

To sweep the actuator, a command is sent and K1 coil is energized which applies power to K2 and K5 coils. Power is also supplied to the probe electronics system. The motor starts rotating in the clockwise direction. After 50 milliseconds, the K1 coil is deenergized. The motor continues in motion until the arm closes the clockwise switch and energizes the second K3 coil. The K3 contacts which determine motor direction are switched, and also K4 coil is energized. The motor reverses direction, and the arm starts to move off the clockwise switch. Because K2 was deenergized by K4 contacts, as the clockwise switch is opened and K4 is deenergized this also deenergizes K5 and the 400-hertz power is off. The motor stops and the power to the probe electronics has also been removed. The actuator is ready for another sweep.

Note that when the clockwise limit switch was closed, capacitor C1 was charged. The opening of the limit switch discharges C1 to the telemetry system for indication that a clockwise sweep has been completed. This event occurs only for clockwise sweeps.

APPENDIX C

COMPONENT ENVIRONMENT TEST OF ACTUATOR FOR SERT II BEAM PROBE

The component environment test consists of the following tests:

(1) Vibration - flat plate for fixture; mass dummy probe in place of actual probe
   - (a) Sine - various gravity inputs at frequencies of 20 to 2000 hertz in each of three axes; different input levels for each axis
   - (b) Random - white noise input at frequencies of 20 to 2000 hertz in each of three axes; different inputs for each axis

(2) Shock - three half-sine pulses of ±30 g's peak amplitude for duration of 8 milliseconds in each of three axes

(3) Thermal vacuum - actual probe installed for this test; component in vacuum tank at pressure of 1×10^{-6} torr or less, subjected to -35° C for 4 hours and operationally tested; then soaked at 50° C for 48 hours; operationally tested twice during the 48-hour period, and again at the end of this period

APPENDIX D

PERFORMANCE ACCEPTANCE TEST OF ACTUATOR FOR SERT II BEAM PROBE

Each performance test consisted of three operating cycles (a cycle consists of one clockwise sweep and one counterclockwise sweep) with these electrical inputs:

(1) Motor input voltage, 115 volts, 400 hertz for all sweeps

(2) dc voltage settings

  (a) 22 volts first cycle

  (b) 28 volts second cycle

  (c) 35 volts third cycle

The following parameters were measured for each sweep (limits are given):

(1) Time of sweep, 49±5 seconds (all actuators)

(2) Motor input current, 60 to 127 milliamperes

(3) Power factor must be greater than 0.9

(4) Telemetry signal for probe position (see fig. 9) must be 1/2 volt after 30 seconds (This measurement is for clockwise sweeps only.)

The postoperational open circuit check consists of resistance check between each pin of the input connector (J1 in fig. 9) and electrical support structure. Resistance must be essentially infinite.

REFERENCES

1. Kerslake, William R.; Byers, David C.; Staggs, John F.: SERT II Experimental Thrustor System. Paper 67-700, AIAA, Sept. 1967.

2. Goldman, Richard G.; Gurski, Guy S.; and Hawersaat, William H.: Description of the SERT II Spacecraft and Mission. Presented at the AIAA Eighth Electric Propulsion Conference, Stanford, Calif., Aug. 31-Sept. 2, 1970.

3. Vernon, Richard H.; and Daley, Howard L.: Emissive Probes for Plasma Potential Measurements on the SERT II Spacecraft. Paper 69-272, AIAA, Mar. 1969.

TABLE I. - TEST PROGRAM: ACTUATOR FOR SERT II BEAM PROBE

Unit	Type of test	Date	Results of test
Experimental	Experimental vibration to determine resonant frequencies and amplification factors.	2/29/68	At certain frequencies, amplification factors too large. Structural reinforcement required.
Experimental	Rerun of above test to check effect of modifications.	4/18/68	Amplification factors acceptable.
Prototype	Acceptance tests as components for prototype spacecraft. Appendix C inputs.	7/18/68 to 7/23/68	Units passed all tests. Ready for prototype spacecraft.
Qualification	Design qualification testing: vibration, shock, and thermal vacuum.	10/25/68 to 12/16/68	Design was qualified for flight.
Prototype	Vibration, shock, and thermal vacuum tests for components on prototype spacecraft.	7/31/68 to 9/25/68	Failure of pin puller on another component created high input to one assembly. Limit switch would not actuate. Method of mounting switches improved.
Flight	Acceptance tests as components for flight spacecraft. Appendix C inputs.	11/5/68 to 11/20/68	Both units acceptable for flight spacecraft.
Backup	Acceptance tests as components for flight spacecraft. Appendix C inputs.	8/14/69 to 9/7/69	Unit acceptable for flight spacecraft.
Flight	Flight acceptance of assembled spacecraft: vibration, shock, and thermal vacuum.	12/8/69 to 12/31/69	Both units ready for launch.
Qualification	Beam probe and transformer modifications required further testing of these units. Actuator used as "test fixture" for vibration and shock tests at qualification levels.	3/5/69 to 3/19/69	Actuator was able to pass all operational tests after this second qualification vibration and shock test.
Prototype	Thermal vacuum endurance test of ion thruster with actuator and probe used to collect thruster data.	10/22/69 to 11/13/69	Actuator performed 1130 cycles of operation in simulated space environment with live thruster. Unit still operating properly at end of test.

TABLE II. - ACTUATOR OPERATIONS OF

SERT II SPACECRAFT

[Launch: February 3, 1970; pin pullers fired February 7, 1970 - both actuators; arms moved out of launch clamp February 7, 1970.]

Actuator 1 (with operating thruster)		Actuator 2	
Date	Sweeps	Date	Sweeps
2/7/70	2	2/7/70	2
2/18/70	6	2/18/70	4
2/19/70	8	3/5/70	2
2/26/70	2	5/29/70	2
2/27/70	2	6/3/70	2
3/2/70	2		
3/5/70	4		
3/12/70	2		
3/13/70	2		
3/19/70	2		
3/20/70	2		
3/27/70	8		
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4/3/70	2		
4/29/70	2		
5/19/70	2		
6/3/70	2		

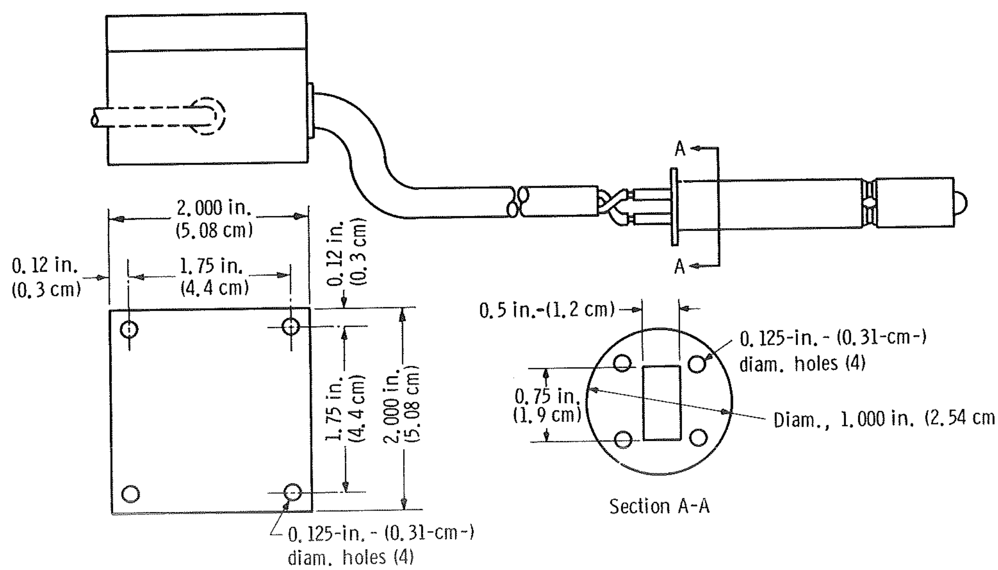


Figure 1. - Beam probe and transformer configuration and interfaces.

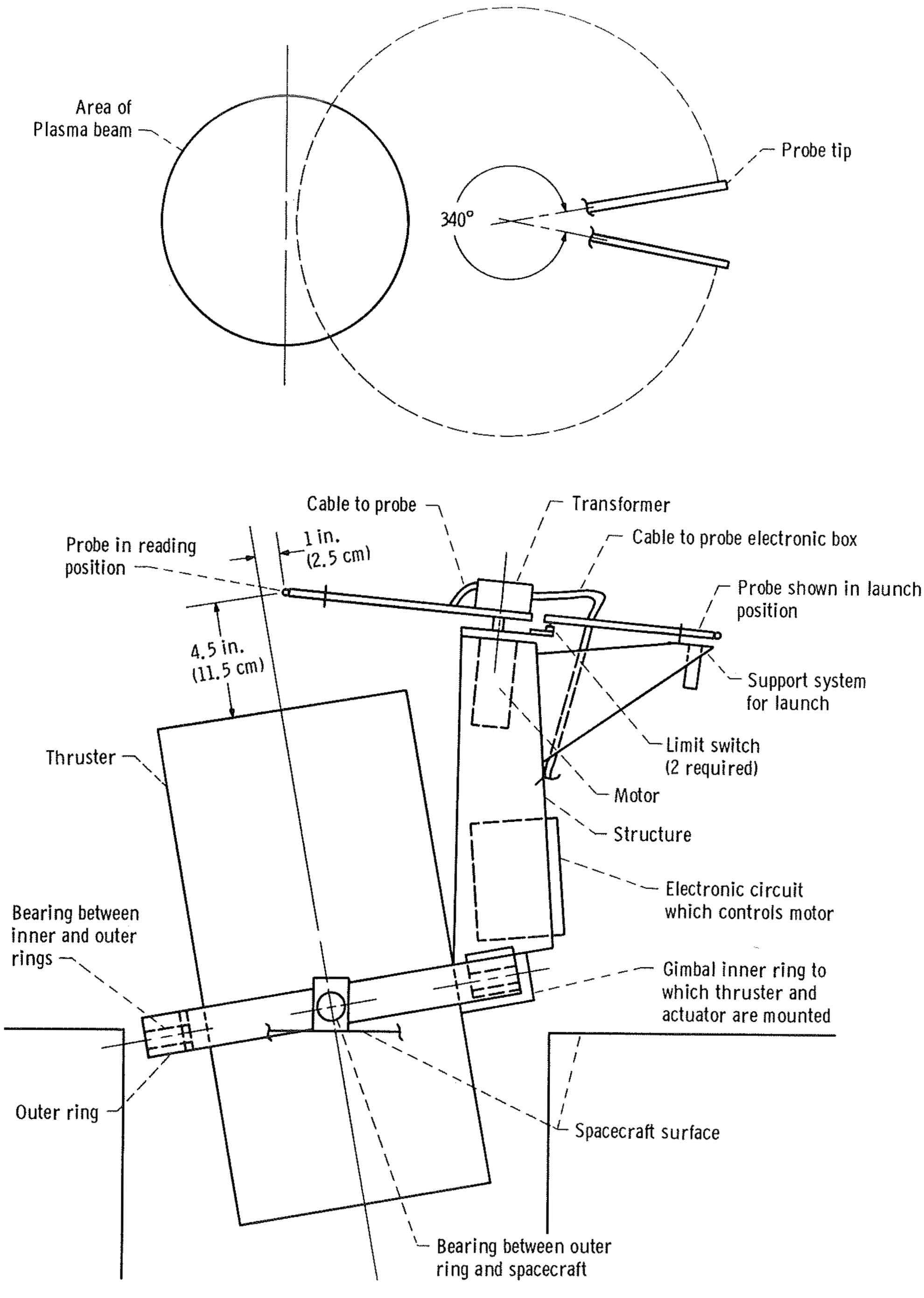
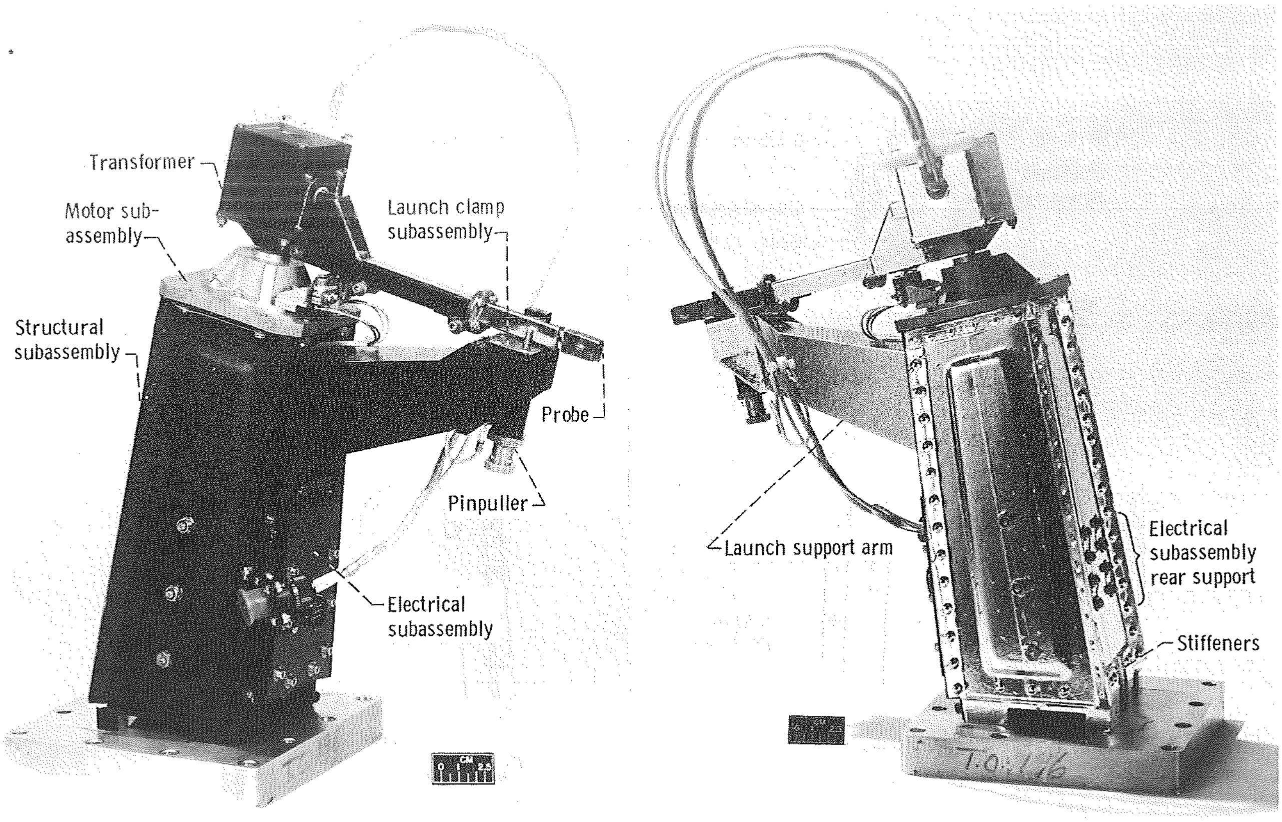


Figure 2. - Conceptual design.



(a) Space side. C-70-558

(b) Sun side. C-70-559

Figure 3. - Actuator assembly.

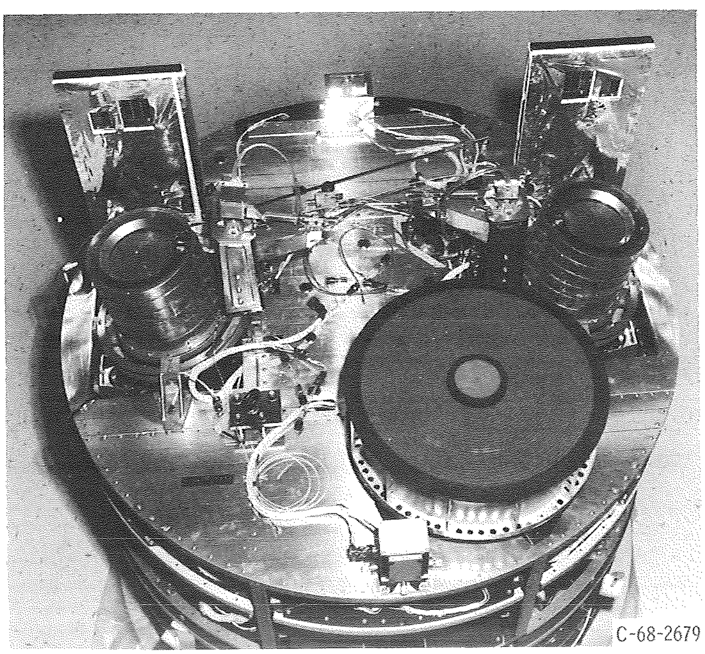


Figure 4. - Location of actuator on SERT II spacecraft.

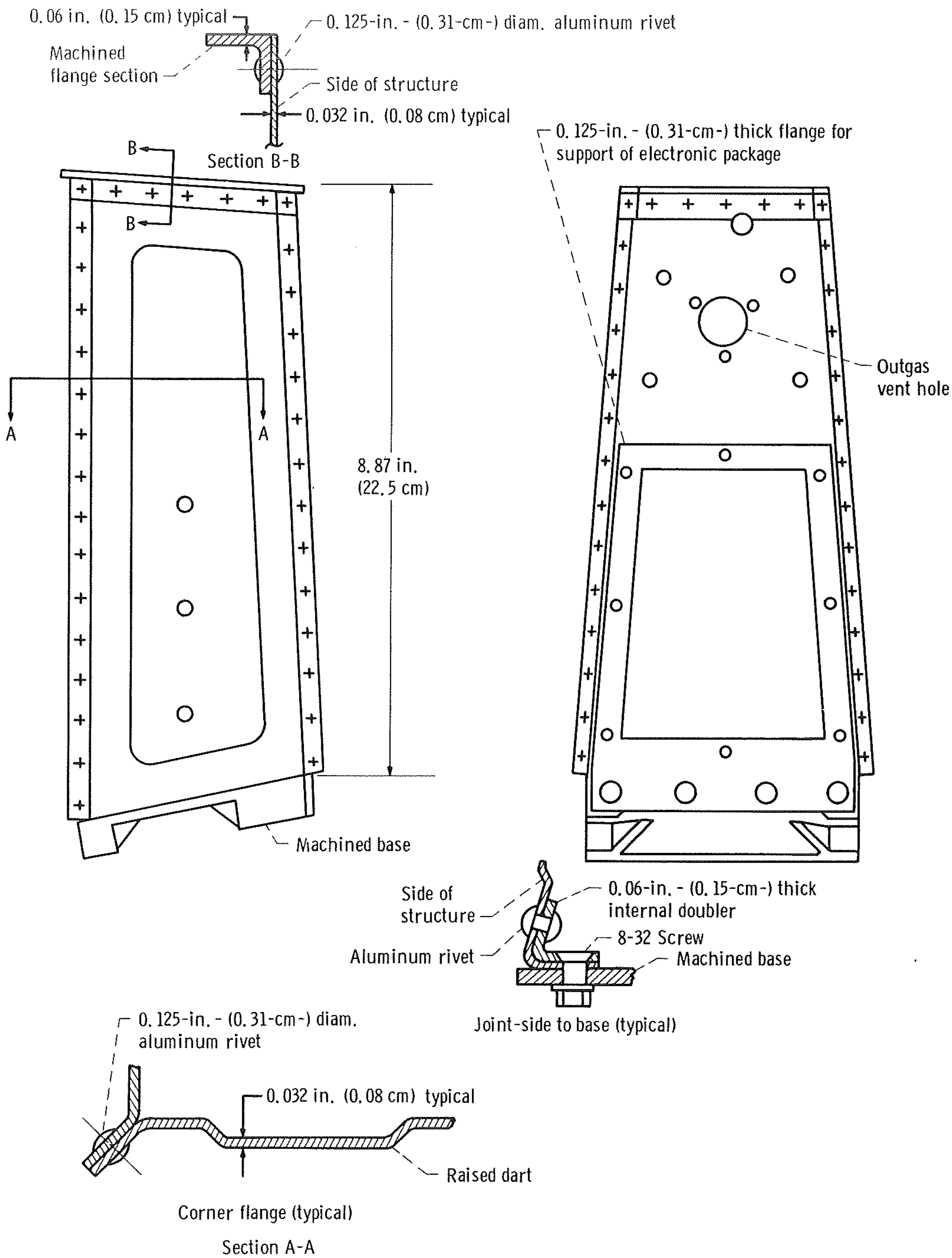


Figure 5. - Beam probe actuator structure.

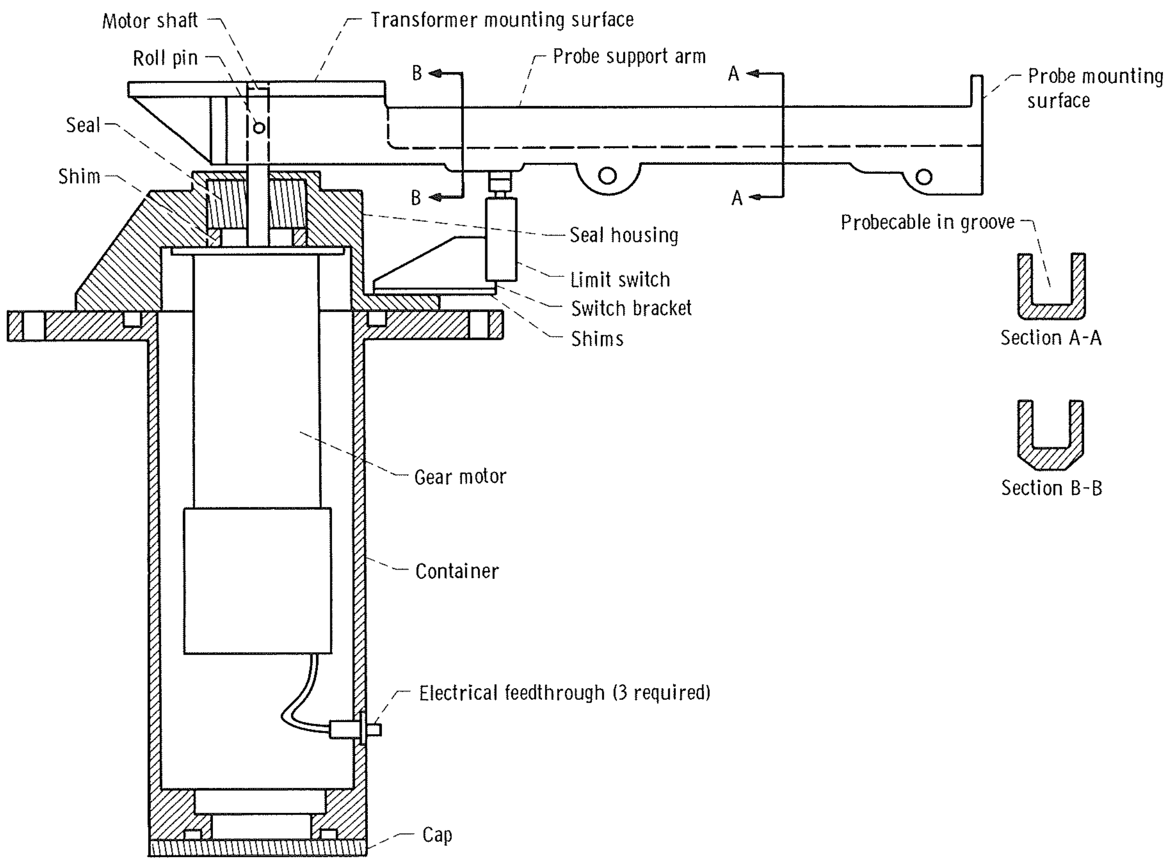


Figure 6. - Beam probe actuator motor subassembly.

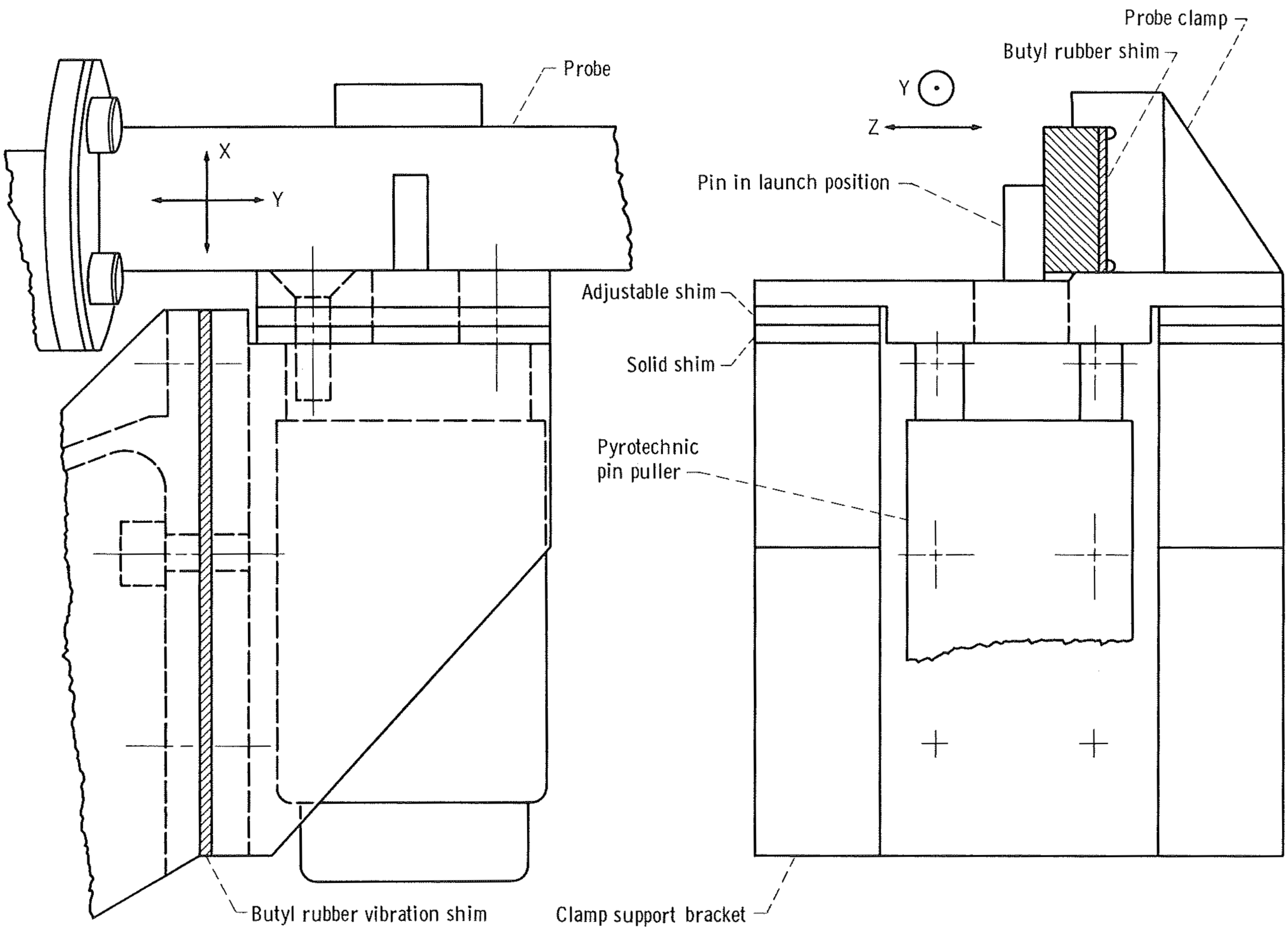


Figure 7. - Beam probe actuator launch clamp.

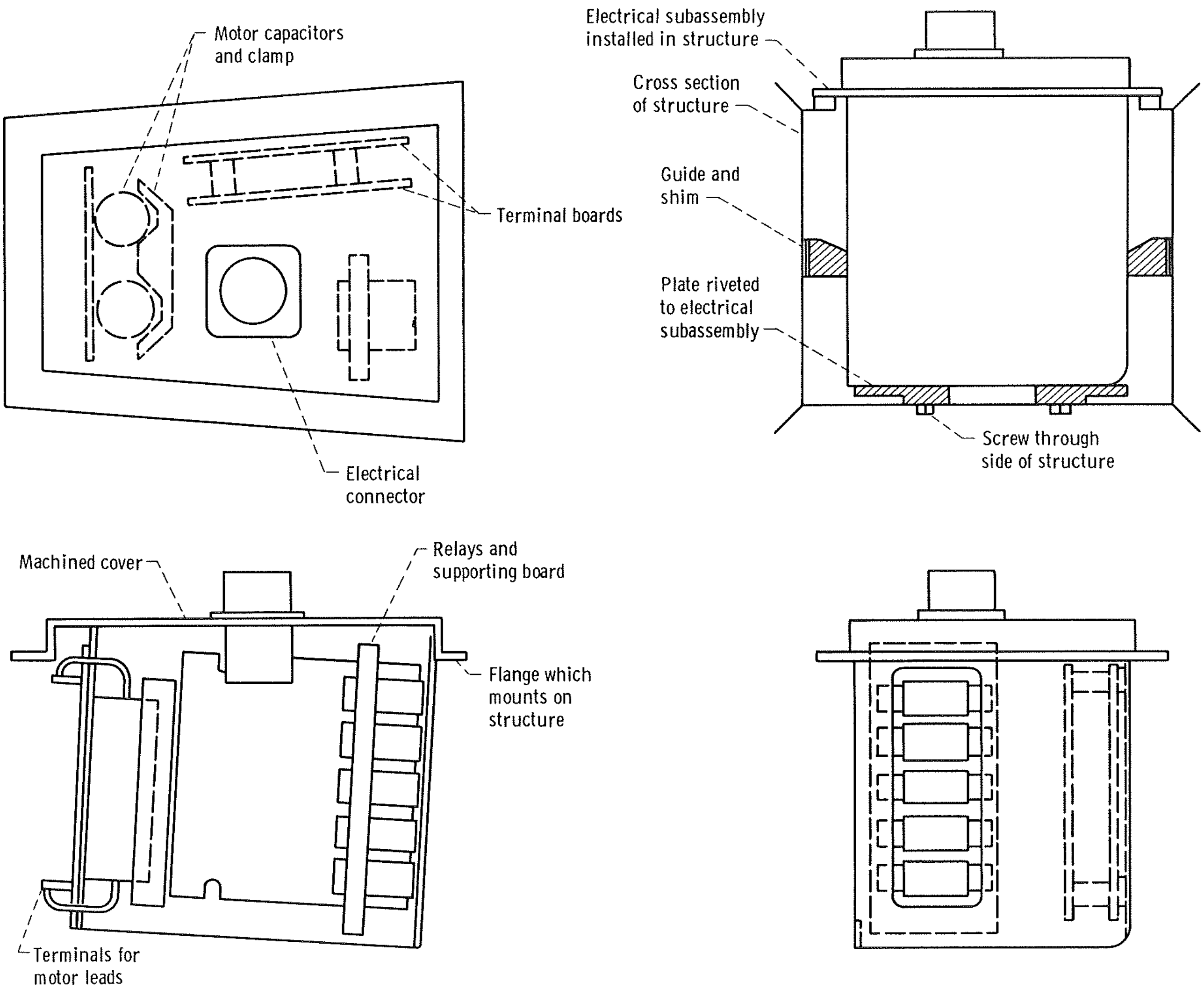
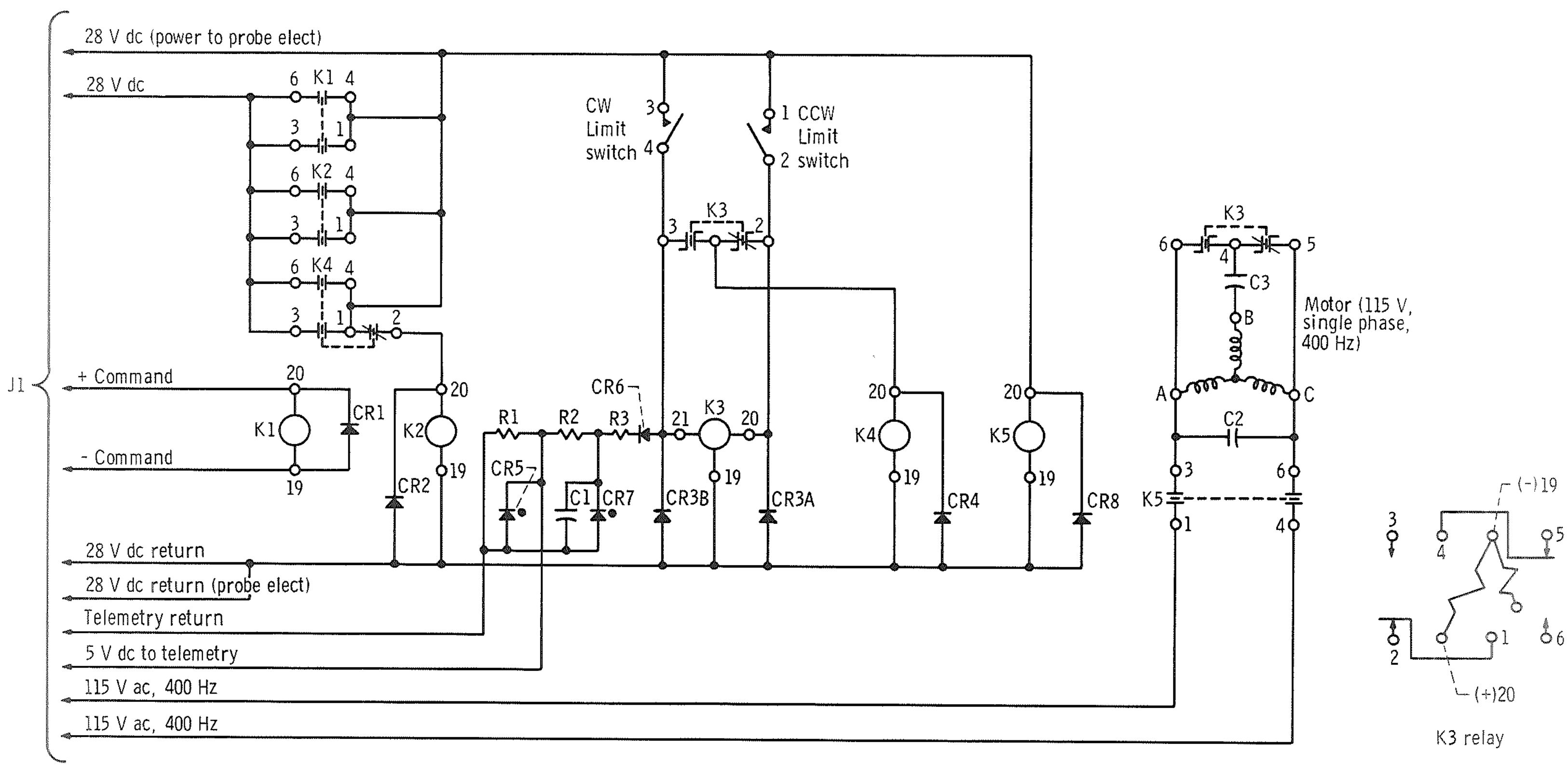


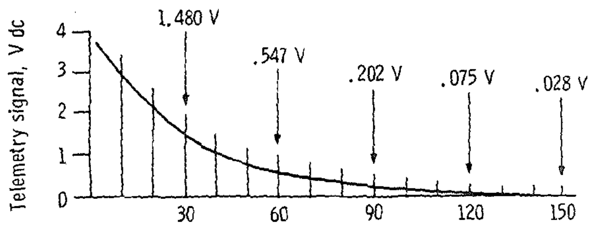
Figure 8. - Electronic subassembly.



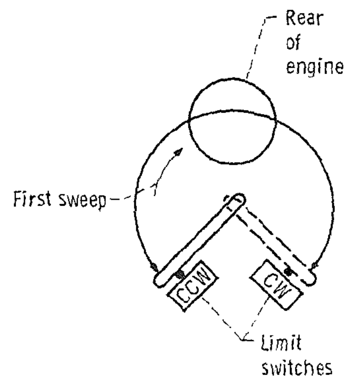
(a) Electrical schematic of beam probe actuator.

NASA-Langley, 1971 — 3 E-5957

Designation	Description	Specification
R1	Resistor, 133K ± 1 percent RNR65C	MIL-R-55182/5
R2	Resistor, 511K ± 1 percent RNR65C	MIL-R-55182/5
R3	Resistor, 1.5K ± 1 percent RNR65C	MIL-R-55182/5
C1	Capacitor, 47MF, 35V dc	LeRC82267-0/2
C2	Capacitor, 0.22MF, 400 working V dc, 10 percent	LeRC82267-0/4
C3	Capacitor, 0.39 MF, 400 working V dc, 10 percent	LeRC82267-0/3
CR1, CR2, CR3A, CR3B, CR4, CR6, CR8	Diode, 1N649	MIL-S-19500/240A
CR5	Diode, Zener, 1N753A, 6.2 V	MIL-S-19500/127c
CR7	Diode, Zener, 1N3027B, 20 V	MIL-S-19500/115D
K-1, K-2, K-4, K-5	Relay, latching	Marshall Space Flight Center 399/52
K-3	Relay, latching,	Marshall Space Flight Center, 399/42
CW and CCW Limit switch	Limit switch	MIL-S-8805/46
J1	Receptacle	NAS1650R-12-B10PN
Motor	Gear motor	MIL-E-5272
Wire	Number 20 with polyimide insulation	MIL-W-81381



Telemetry output - clockwise limit switch actuated



Probe sweep orientation diagram (First sweep is defined as clockwise when looking at rear (exhaust) of engine)

(b) Telemetry output signal and component description.

Figure 9. - Electrical operation of actuator.

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